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THESIS

EVALUATION OF THE LONGITUDINAL STATIC STABILITY OF THE US-2A AIRCRAFT

by

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Evaluation of the Longitudinal Static

Stability of the US-2A Aircraft

by

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ABSTRACT

An evaluation of the longitudinal static stability of a Navy US-2A aircraft was made. The aircraft was equipped with a Data Acquisition System capable of sensing and recording aerodynamic data other than that given by the standard cockpit instruments. Recorded data were displayed graphically and the Neutral Points for stick-free stability and stick-fixed stability were determined. Stick-free static margin and stick-fixed static margin were determined and evaluated.

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AC	Aerodynamic Center	
α_0	Angle of attack for zero lift	degrees
c	Wing chord	in
Ce	Elevator chord	in
C. G.	Center of gravity	
C_{h_α}	Hinge moment coefficient derivative for angle of attack	
C_{h_δ}	Hinge moment coefficient derivative for elevator deflection	
$C_{h_{\delta_t}}$	Hinge moment coefficient derivative for elevator tab	
C_{ho}	Hinge moment coefficient at zero lift	
C_L	Lift coefficient = L/qS	
C_m	Moment coefficient = m/qSc	
Cm_δ	Moment coefficient for elevator deflection = $\dfrac{d(Cm)}{d(\delta e)}$	
δ_e	Elevator deflection	degrees
δ_{e_0}	Elevator deflection at zero lift	degrees
δ_t	Tab deflection	degrees
Fe	Stick force	lbs
G	Elevator gearing	ft
i_w	Wing incidence angle	degrees
i_t	Tail incidence angle	degrees

L	Lift	lbs
L_t	Lift due tail	lbs
l_t	Distance between L_t and C. G.	in
MAC	Mean aerodynamic chord	
No	Stick-Free Neutral Point	
No$'$	Stick-fixed neutral point	
η_t	Tail efficiency	
q	Dynamic pressure ($\frac{1}{2}\rho V^2$)	lbs/ft
Se	Elevator area	ft
Vs	Stall speed	kts
W	Weight	lbs
Xa	Distance between C. G. and AC	in
$X_{C.G.}$	Position of C. G. (% MAC)	

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I. INTRODUCTION

In the flight mechanics curricula at the Naval Postgraduate School the sequence of courses in Flight Evaluation Techniques is supported by a US-2A aircraft equipped with a Data Acquisition System capable of sensing and recording aerodynamic data other than that given by the standard cockpit flight and engine instruments.

The US-2A is a utility conversion of the S-2A aircraft and is used primarily for pilot training and administrative support. Since 1952 over one thousand of these aircraft have been produced in various configurations. Inasmuch as there were no stability data available on the US-2A modification, the static longitudinal stability of this aircraft was analyzed. In addition, further evaluation of the Data Acquisition System installation, which had had no prior usage, was made.

The investigation was performed at the Naval Postgraduate School, Monterey, California, during the period July 1969 through March 1970, and involved fourteen flights with a total of twenty-seven hours of flight time.

II. AIRCRAFT DESCRIPTION

The US-2A aircraft is a four-place, twin-engine, high-wing monoplane manufactured by the Grumman Aircraft Engineering Corporation of Bethpage, New York. The S-2A aircraft was originally designed as a carrier-based anti-submarine aircraft. The US-2A is a utility version of the S-2A. All anti-submarine warfare equipment has been removed. The overall dimensions of the aircraft are shown in Figure 1. The aircraft systems essential to this investigation were the power plant, electrical system, and flight control system. They are described below.

The aircraft is powered by two Wright R1820-82A nine-cylinder reciprocating engines, capable of producing 1525 brake horsepower at an engine speed of 2800 revolutions per minute and a manifold pressure of 56.5 inches of mercury at sea level on a standard day. Each engine is equipped with a three-bladed, variable pitch, full feathering propeller.

The aircraft's electrical power supply systems consist of a 24-volt direct current, monitored bus system and a 115-volt, fixed frequency, alternating current system. The dc system is powered by two 300-ampere, engine-driven generators regulated to 27.7 volts and by a 24 volt/36 ampere hour storage battery. The two dc generators are connected in parallel and direct current is distributed through a system of busses. The ac system consists

of main, a standby, and a pilot's instrument inverter. The inverters are 115 volt, three phase, 400 Hertz motor-generator combinations that operate on direct current from the dc bus system.

The flight control system of the aircraft consists of mechanical, hydraulic, and electrically actuated control surfaces.

Conventional elevators provide longitudinal control. The elevators are mechanically actuated by the pilot's or co-pilot's control wheel. An electrically actuated trim tab and a geared tab are installed on both the right and left elevator.

Lateral control is accomplished by ailerons and two spoilers in each outer wing panel. The ailerons are mechanically actuated by the pilot's or co-pilot's control wheel. The spoilers are mechanically linked with the ailerons.

The directional control system consists of a conventional rudder, actuated by the pilot's or co-pilot's rudder pedals. A rudder trimmer which can be actuated either electrically or hydraulically is available.

The pilot and co-pilot are provided with standard flight and engine instruments.

III. DATA ACQUISITION SYSTEM

The Data Acquisition System is designed to provide a means of accurately measuring and continuously recording those parameters which are necessary for stability and control flight testing but which are not presented on the aircraft's instrument panel. More specifically, these parameters are control forces, primary surface control position, normal acceleration, angle of attack, angle of sideslip, pitch rate, roll rate, and yaw rate.

A. DATA RECORDING

The recording device is an Ampex Series 800 Magnetic Tape Recorder. It is shown in Figure 2 and consists of the following components:

1. A shock-mounted tape transport mechanism which consists of a closed loop tape drive system and a plug-in type head assembly.

2. A shock-mounted cabinet designed to accommodate seven plug-in record amplifiers, either of the frequency modulation or amplitude/pulse width modulation type in any desired combination.

3. A shock-mounted electronics power supply which furnishes regulated voltages for the record amplifiers and the tape transport record head.

4. A shock-mounted capstan power unit which consists of a 400 Hertz power supply, a power amplifier, and a 60 Hertz

oscillator which are used to supply frequency stabilized 60 Hertz power to the capstan motor in the tape transport.

5. A remote control unit which controls the distribution of power to all components in the system by means of a "power" switch, and which controls tape transport operation and the amplifier record relays by means of a "record" switch.

6. A test unit which provides a pre-operational check-out of the recorder by systematically examining selected system voltage, the carrier head current in the FM amplifiers, the bias level of the oscillator in the electronics power supply, and signal head current of the AM record amplifiers.

The recorder's primary power requirements of 28 volt dc and 115 volt, 400 Hertz are provided by the aircraft's dc bus and main inverter, respectively, through a power terminal board. This regulated power is then distributed to the various components of the tape recorder through a system of plug-in interconnecting cables. Circuit protection is provided by a five ampere dc circuit breaker marked RCDR located on the co-pilot's circuit breaker panel, and a five ampere fuse marked RCDR located on the ac distribution box.

The recorder utilizes a 1/2 inch tape operating at a speed of fifteen inches per second. Six channels for high frequency aerodynamics data and one channel for wide band direct voice recording

are provided.

The electrical signals that are generated by the various transducers used to sense aerodynamic data are modified so that they provide a dc voltage amplitude within the range $\pm$ 1.4 volts. The electrical signals are then transmitted to a signal terminal board. From the signal terminal board the input signals are transmitted via co-axial cables to connectors located at the rear of the record amplifier cabinet. The signal terminal board and associated co-axial cables provide for thirteen signals. The cables may be connected to any of the six FM record amplifiers thus providing flexibility in which aerodynamic parameters are to be recorded.

Voice transmissions are obtained from the pilot's intercommunication system microphone and transmitted to the AM record amplifier via the signal terminal board and corresponding co-axial cable.

B. MEASUREMENT OF CONTROL FORCES

A control wheel force transmitter, shown in Figure 3, is used to measure elevator and aileron control forces. This device employs two cantilever beams which are deflected fore and aft as force is applied to move the elevator control surface. Eight 120 ohm resistance strain gages are bonded to the cantilever beams forming two Wheatstone bridges which provide voltage amplitude signals that are related to the forces applied.

To measure rudder pedal forces, four 120-ohm resistance strain gages were bonded to the co-pilot's rudder push-rods in the form of a Wheatstone bridge to provide a voltage amplitude signal which is related to the difference in forces applied to the co-pilot's rudder pedals.

The control wheel force transmitter will measure elevator control forces of up to twenty pounds of push and fifty pounds of pull and maximum aileron forces of fifty pounds. The rudder force measurement installation will measure up to 180 pounds of directional control forces.

C. CONTROL SURFACE POSITION

To measure control surface position, transducers are connected to actuators which have a linear displacement with respect to the angular position of the control surface.

To measure elevator angular deflection a sliding arm potentiometer is attached to the pushrod between the elevator control sector and the bellcrank assembly.

The angular position of the aileron is measured by a transducer connected to the aileron control rod which extends across the aircraft fuselage. Another transducer is connected to the right rudder control cable to determine the angular position of the rudder.

A five volt dc power supply, activated by the aircraft's 115 volt, 400 Hertz main inverter, provides a power source for the three control surface position transducers.

D. CALIBRATION

To accomplish this particular investigation it was necessary for the Data Acquisition System to measure and record control stick forces and elevator deflections. To ensure that reliable data was obtained during the flights and to provide calibration curves that would relate a given control input to a signal output, it was necessary to calibrate the system.

Preliminary test flights indicated that control forces of up to ten pounds of push and ten pounds of pull and elevator deflections of no more than $\pm$ 5 degrees would be required.

To calibrate the control wheel, the system was adjusted for zero output with no force applied. Forces of known magnitude were then applied in two pound increments up to twelve pounds, and the resulting signal output recorded. The resulting data is given in Table I and the calibration curve is shown in Figure 4. The calibration curve relating the angular position of the elevator to signal output was obtained by displacing the trailing edge and recording the resulting signal. The elevators were locked in the neutral position and the output adjusted for a zero reading. Using an inclinometer situated so that its center coincided with the elevator hinge line, the trailing edge was displaced in increments

of five degrees, over a range of fifteen degrees down, to twenty-five degrees up and the output recorded. The resulting data is given in Table II and the calibration curve is shown in Figure 5.

IV. FLIGHT PROCEDURES

Prior to flight, the NATOPS Flight Manual was consulted to determine the trim speed to be used and the speed ranges to be covered. In flight testing, trim speeds are, for standardization purposes, multiples of stalling speeds. The US-2A stalls at 80 knots and a trim speed of twice Vs or 160 knots was used for these tests. Once the trim speed and configuration of the test was set, efforts were made to keep power constant as variations from the test altitude occurred. For reciprocating engine aircraft, constant manifold pressure suffices over a $\pm$ 2000 foot altitude range. It was required that these tests be run at several different center of gravity locations. Since there is no mechanical means to alter the center of gravity in the US-2A aircraft, this was accomplished by having two crew members position themselves at station 302 for an aft center of gravity, station 167 for a midships center of gravity, and stations 120 and 140 for a forward center of gravity.

Before starting, a preflight of the Data Acquisition System was made ensuring that each component was operating properly. The output voltages from the control stick force transmitter and elevator deflection potentiometer were checked and set at zero if necessary.

Normal start, takeoff, and taxi procedures were followed. After takeoff, a climb to an altitude of 5000 feet was made. After

leveling off, the pilot carefully trimmed the aircraft at 160 knots in cruise configuration with the two crew members in the aft center of gravity position. The power switch and tape record switch were turned to the "on" position.

The pilot then increased the speed by ten knots by pushing the stick forward. Once the speed was stabilized the pilot in the right seat took control of the aircraft and applied the required force to the control wheel force transmitter to maintain that airspeed. He held this for approximately ten seconds, released the control wheel force transmitter and returned control to the pilot in the left seat. Therefore, only the force necessary to hold the airspeed was applied to the control force transmitter and only during the desired periods. Using this method, the required force readings were clearly defined on the playback of the tape as shown in Figure 6.

At the same time, on another record channel, the elevator deflection was recorded. A new speed was then attained and the force and deflection again recorded as in the previous run. This was continued in increments of ten knots above and below trim speed over a range from 130 knots to 200 knots. During these runs, changes of altitude inevitably occurred. To prevent too much altitude from being gained or lost, it was necessary to alternate taking high and low speed points. Using this method a change of

no more than 1500 feet occurred during any one run and the runs were alternated around the test altitude of 5000 feet.

When the range of airspeeds had been covered in this configuration, the center of gravity of the aircraft was shifted by moving the crewmen to a midships position and a similar set of runs made.

Finally runs were made with the crewmen in the forward center of gravity position.

At the end of these tests, runs had been made at three different center of gravity positions and the stick force and elevator deflections recorded for each.

V. DATA REDUCTION

To analyze the data recorded while in flight, the tape from the Data Acquisition System was played back on a ground-based Ampex FR1100 Magnetic Tape Record/Reproduce unit. Output from the Ampex FR1100 was put into a Brush Electronics Corporation type BL-817 Recorder, which displayed the data graphically.

Using the calibration curves obtained from the known inputs, it was possible to determine the control stick force and elevator deflection for each test airspeed.

To use these data to arrive at the longitudinal stability of the aircraft it was necessary to consider the following theory of aircraft stability and control.

Reference 1 states that the static longitudinal stability of an airplane is defined as the initial tendency displayed by an airplane following displacement in the pitching plane from some equilibrium (trim) condition. When an airplane is so displaced, pitching moments may or may not occur. In the stable case, the moments will cause the airplane to return toward the original equilibrium condition. An unstable airplane will tend to diverge further from the equilibrium condition. A neutrally stable airplane will remain at the condition to which it was displaced.

Since the elevators of a conventional airplane are moveable, changes in angle of attack cause the elevators to float with the relative wind from the original trim position. As a result the tail

lift produced by the increase in angle of attack is different than that which would have resulted from the same change in angle of attack had the floating tendencies been eliminated by restraining the stick in the original trim condition. This variation in tail lift with the stick fixed and with the stick free to float causes the aerodynamic moments to vary. Thus we have the terms Stick-Free stability and Stick-Fixed stability. In flight, we cannot measure these two types of stability directly, but we must measure quantities which are related. These related quantities are the longitudinal stick force (Fe) and elevator position (δe). The variation of longitudinal stick force and elevator position with velocity about a particular trim speed constitute stick-force and elevator-position stability in the cockpit which, in turn, are related to the stick-free and stick-fixed stabilities, respectively, of the airplane. In order to see how these are related, consider the forces on an aircraft in flight (see Figure 7). In trimmed level flight, the net lift must equal the weight of the airplane and the sum of the applied moments must be zero. The airplane lift minus the tail lift is considered to act through the aerodynamic center. The aircraft has longitudinal stick-fixed stability if the a. c. is aft of the center of gravity. If the net lift equals the weight, then

$$L - L_t = W$$

For the sum of the moments to equal zero, the following must hold:

$$L_t \times l_t = L \times Xa$$

The tail lift is controlled by variations in elevator deflection as necessary to establish zero moment. As speed is increased the angle of attack required to maintain the same lift decreases. This causes an increase in the downward directed lift of the tail. Thus to establish equilibrium at a higher speed, on an airplane with stick-fixed stability, the stick must be moved forward, causing the elevator's trailing edge to be moved downward. This is normal and stable elevator position variation. At the same time, if the "push" stick force increases with increasing velocity, i.e., in order to move the elevator trailing edge downward, then stick-force stability also exists.

In Ref. 2 it is shown that for an aircraft to be stable the slope of Cm versus C_L must be negative as shown in Figure 8. Thus $\frac{d(Cm)}{d(C_L)}$ is called the stability criterion and the stability equation for an aircraft is shown to be the sum of the stability contributions of the wing, tail, fuselage and power plant respectively.

It can be shown that the equation for stick force per unit dynamic pressure is given as,

$$Fe/q = K(A + C_{h_{\delta_t}} \delta_t) - K \frac{C_{h_\alpha}}{Cm_\delta} \left(\frac{dCm}{dC_L}\right)_{Free} C_L$$

Where $K = -G\, S_e\, c_e\, \eta_t$

AND $A = C_{h_0} + C_{h_\alpha}(\alpha_0 - i_w + i_t) + C_{h_\delta}\, \delta_{e_0}$

Taking the derivation with respect to C_L,

$$\frac{d(Fe/q)}{d(C_L)} = -K \frac{C_{h_\delta}}{C_{m_\delta}} \left(\frac{dC_m}{dC_L}\right)_{FREE}$$

Thus, the slope of Fe/q versus C_L is a direct function of the stability criterion $(dCm/dC_L)_{FREE}$. In reducing flight data, curves of Fe/q versus C_L were plotted. The slope of these curves was then plotted versus center of gravity position, and the center of gravity position for zero $\frac{d(Fe/q)}{d(C_L)}$ interpolated or extrapolated. This center of gravity position was the Stick-Free Neutral Point for the lift coefficient at which the slopes were taken.

The Stick-Fixed Neutral Point was computed in a similar way. Curves of δe versus C_L were plotted for various centers of gravity. The slopes of these curves were taken and a plot of $\frac{d(\delta e)}{d(C_L)}$ versus center of gravity made. The Stick-Fixed Neutral Point was where the value of $\frac{d(\delta e)}{d(C_L)}$ is equal to zero.

VI. DISCUSSION OF RESULTS

To find the center of gravity location in percent mean aerodynamic chord (MAC), a plot of moment versus fuel weight was made as shown in Figure 9. This accounted for two pilots but no crew members. As the test flights were made, the moment caused by the crew members at their respective stations was added. Entering the Center of Gravity Table in Ref. 3 with the value of total moment, the center of gravity positions for the aft, midship, and forward crew positions were found to be 24.9% MAC, 22.9% MAC, and 21.3% MAC respectively.

For ease in converting flight test airspeeds, a curve of C_L versus Indicated Airspeed was made and is shown in Figure 10.

Following these preliminary calculations, it was possible to plot the recorded flight data for stick-free stability and stick-fixed stability, respectively.

A. STICK-FREE STABILITY

Graphs of Fe/q versus C_L were made for center of gravity positions 24.9% MAC, 22.9% MAC, and 21.3% MAC, and are shown in Figures 11 through 13, respectively. These curves indicate a positive stick-free stability for this aircraft, in that each curve has a negative slope. This indicates that as the nose of the aircraft is lowered, thus increasing airspeed and decreasing C_L, a moment

is developed that tends to raise the nose and restore the aircraft to the trim condition. To hold this higher airspeed, a push force is required to counteract this stabilizing moment. This push force is the recorded stick force. The farther away the airspeed is from the trim airspeed, the greater is the required stick force. For airspeeds lower than trim airspeed a pull force is required to counteract the stabilizing moment that tends to push the nose back to trim condition.

The value of the slope of the curve was taken graphically at C_L values of .3, .4, .5, and .6. This was then plotted versus center of gravity (C. G.) position giving Figure 14. By extrapolating each curve until it crossed the abscissa, the Neutral Point (No) for the particular value of C_L was found. The Neutral Point is the C. G. location at which the change in Fe/q with respect to C_L is zero.

The stick-free Neutral Point was then plotted versus C_L as shown in Figure 15. This shows that for normal cruise airspeeds the Neutral Point ranges from 27.6% MAC to 31.9% MAC. Ref. 2 shows that the stability equation can be reduced to,

$$\frac{d\ Cm}{d\ C_L} = \frac{X}{C} = X_{C.G.} - N_o$$

The right hand side of this equation is the negative of the static margin. Static margin was computed and is shown in Figure 16. For the aft center of gravity position, a static margin range of

from 2.7% MAC to 7.0% MAC existed. The midships center of gravity position indicated a static margin range of from 4.7% MAC to 9.0% MAC. At the forward center of gravity position a range of from 6.3% MAC to 10.5% MAC was present. This demonstrates that for the cruise speed range of the aircraft and over the extent of movement of the center of gravity, a safe margin for positive stick-free longitudinal stability exists. Table III contains the data for stick-free stability.

B. STICK-FIXED STABILITY

Graphs of δ_e versus C_L were plotted for the three center of gravity locations and are shown in Figures 17 through 19. From these it can be seen that the aircraft exhibits positive stick-fixed stability. The elevator must be deflected trailing edge down to maintain airspeeds higher than trim airspeed. This indicates a nose up restoring moment if the nose is deflected downward. For airspeeds lower than trim airspeed a nose down restoring moment tends to return the nose of the aircraft to the trim position.

As in the case for stick-free stability the slopes of these curves were taken at C_L values of .3, .4., .5, and .6. A graph was then plotted of $\frac{d(\delta_e)}{d(C_L)}$ versus C.G. position. See Figure 20. Extrapolation found the points where there was no change in δ_e with respect to C_L. These were the stick-fixed Neutral Points (No'). Figure 21 shows that stick-fixed Neutral Point varies from

28.3% MAC to 32.5% MAC over the range of cruise airspeeds.

The stick-fixed static margin was computed in a manner similar to that for the stick-free static margin. Figure 22 shows that for the aft center of gravity position, a static margin range of from 3.4% MAC to 7.8% MAC existed. The midships center of gravity position showed a range of from 5.4% MAC to 9.9% MAC. The forward center of gravity position indicated a range of from 7.0% MAC to 11.4% MAC over the range of test airspeeds. As in the case of stick-free stability, there exists a safe margin of stick-fixed stability over the range of airspeeds considered. All data for stick-fixed stability is shown in Table IV.

Comparing Figures 15 and 21, it can be seen that the stick-free Neutral Point lies ahead of the stick-fixed Neutral Point. Thus static margin for stick-fixed stability is greater than that for stick-free stability. This is verified by comparing Figures 16 and 22.

VII. CONCLUSIONS

The US-2A exhibited positive longitudinal static stability for both the Stick-Fixed and Stick-Free case throughout the velocity range and center of gravity excursions of the tests. This is indicated by the fact that the Neutral Point remained aft of the center of gravity in all of the tests. Inasmuch as the distance between the center of gravity and the Neutral Point (Static Margin) is a very good approximation of the static stability, Static Margin being approximately equal to the negative of the rate of change of pitching moment coefficient with lift coefficient, it can be stated that $\frac{d(Cm)}{d(C_L)}$ is negative (positive stability) throughout the range of these tests.

The Neutral Point for Stick-Free stability varied linearly from 27.6% MAC for a Lift Coefficient of 0.3 to 31.9% MAC for a Lift Coefficient of 0.6. This may also be expressed as a change in Stick-Free Neutral Point from 31.9% MAC to 27.6% MAC as velocity increased from 138 knots to 192 knots. This forward movement of the Neutral Point results in a decrease in Static Margin which, in turn, results in a decrease in Stick-Free stability as airspeed is increased.

For the Stick-Fixed case, the Neutral Point varied linearly from 28.3% MAC for a Lift Coefficient of .3 to 32.5% MAC for a Lift Coefficient of .6. As in the Stick-Free case, this forward

movement of the Neutral Point indicates a reduction in Stick-Fixed stability with increasing airspeed. However, as expected, the Stick-Fixed Static Margin is slightly greater than the Stick-Free Static Margin.

The Neutral Point in these tests never moved forward of a point 2.7% MAC aft of the aftmost center of gravity location for the Stick-Free case and 3.4% MAC aft of the aftmost center of gravity location for the Stick-Fixed case. This indicates that even though the longitudinal static stability decreased with increasing velocity, the aircraft maintained sufficient stability even in the most extreme case.

TABLE I

ELEVATOR POSITION CALIBRATION

TRAILING EDGE UP		TRAILING EDGE DOWN	
Position Degrees	Signal Output Volts, dc(Positive)	Position Degrees	Signal Output Volts, dc(Negative)
0	0.000	0	0.000
5	.318	5	.222
10	.540	10	.443
15	.840	15	.690
20	1.112	18	.851
25	1.310		

TABLE II

ELEVATOR CONTROL WHEEL FORCE CALIBRATION

PUSH		PULL	
Force Pounds	Signal Output Volts, dc(Negative)	Force Pounds	Signal Output Volts, dc(Positive)
2	.0744	2	.0590
4	.1326	4	.1127
6	.1850	6	.1724
8	.2558	8	.2515
10	.3159	10	.3142
12	.3431	12	.3611

TABLE III

STICK-FREE STABILITY DATA

				Run I		Run II		Run III	
IAS	C_L	Ve	q	Fe	Fe/q	Fe	Fe/q	Fe	Fe/q
130	.65	134.0	60.78	7.6	.125	10.0	.165	18.4	.303
140	.58	144.1	70.29	6.0	.085	6.0	.085	10.9	.155
150	.50	154.6	80.90	3.4	.042	3.4	.042	5.0	.062
160	.44	165.0	92.16	0.0	.000	0.0	.000	0.0	.000
170	.38	175.0	103.66	2.0	.020	4.8	.046	5.2	.050
180	.35	185.0	115.85	3.2	.028	6.5	.056	5.7	.049
190	.31	195.0	128.71	5.0	.039	9.8	.076	11.5	.089
200	.28	205.0	142.25	8.0	.056	11.8	.083	15.0	.105
				$\frac{d(Fe/q)}{d(C_L)}$	Static Margin	$\frac{d(Fe/q)}{d(C_L)}$	Static Margin	$\frac{d(Fe/q)}{d(C_L)}$	Static Margin
	.30			.20	2.7	.385	4.7	.526	6.3
	.40			.35	4.1	.524	6.1	.714	7.7
	.50			.53	5.2	.830	7.2	1.010	8.8
	.60			.67	7.0	.923	9.0	1.140	10.6

TABLE IV

STICK-FIXED STABILITY DATA

		Run I		Run II		Run III	
IAS	C_L	Volt	$\Delta\delta_e$	Volt	$\Delta\delta_e$	Volt	$\Delta\delta_e$
130	.65	.035	.90	.050	1.1	.070	1.6
140	.58	.025	.60	.040	.9	.050	1.2
150	.50	.010	.20	.015	.3	.022	.5
160	.44	Trim	Trim	Trim	Trim	Trim	Trim
170	.38	.010	.15	.020	.4	.020	.4
180	.35	.015	.30	.020	.4	.030	.6
190	.31	.015	.30	.025	.6	.040	.9
200	.28	.020	.40	.030	.7	.045	1.0
		$\frac{d(\delta_e)}{d(C_L)}$	Static Margin	$\frac{d(\delta_e)}{d(C_L)}$	Static Margin	$\frac{d(\delta_e)}{d(C_L)}$	Static Margin
	.30	1.5	3.4	2.3	5.4	3.3	7.0
	.40	2.5	5.0	3.3	7.0	4.5	8.6
	.50	3.3	6.5	4.2	8.5	5.4	10.1
	.60	4.0	7.6	5.0	9.6	6.0	11.2

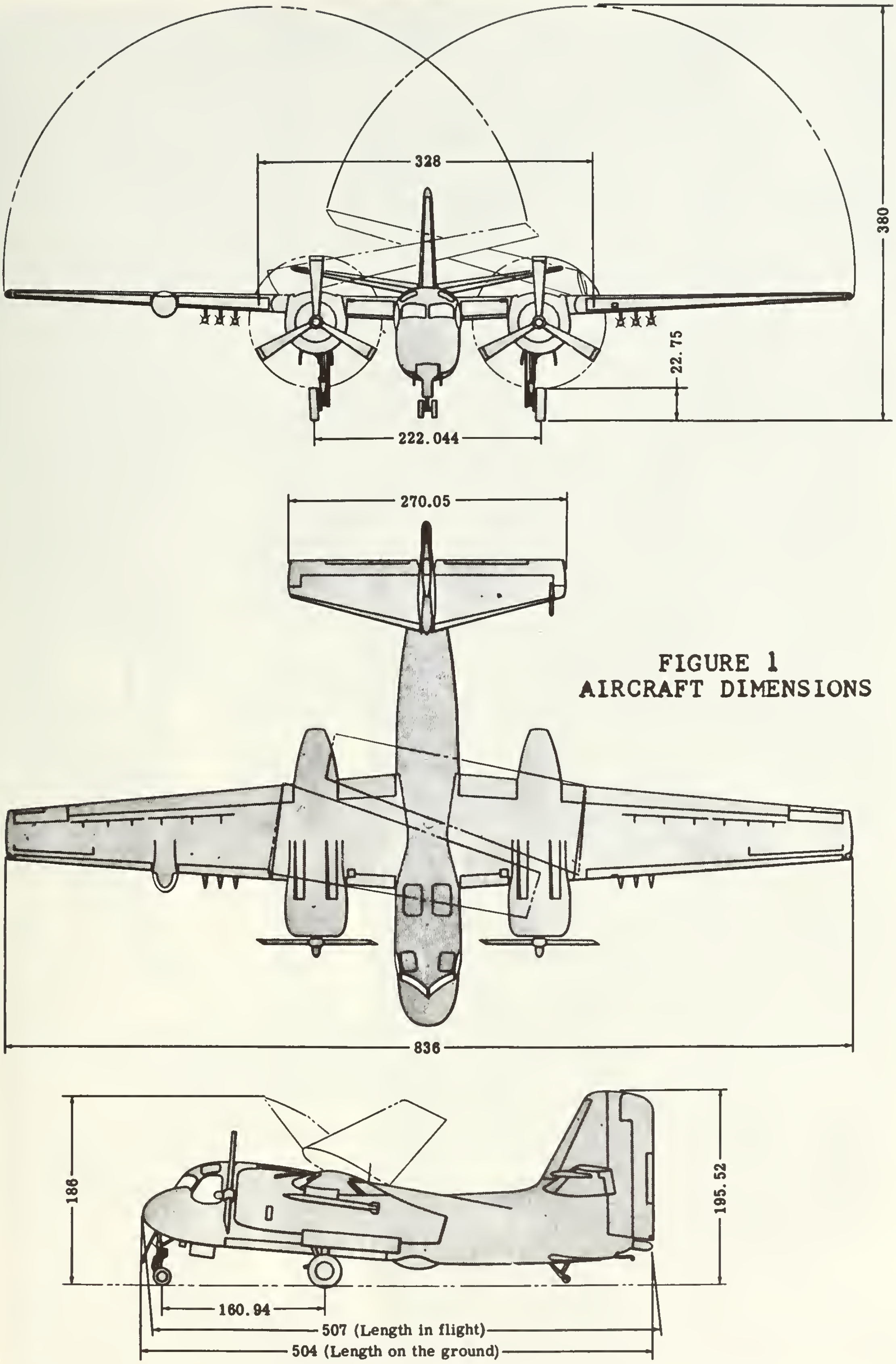

FIGURE 1
AIRCRAFT DIMENSIONS

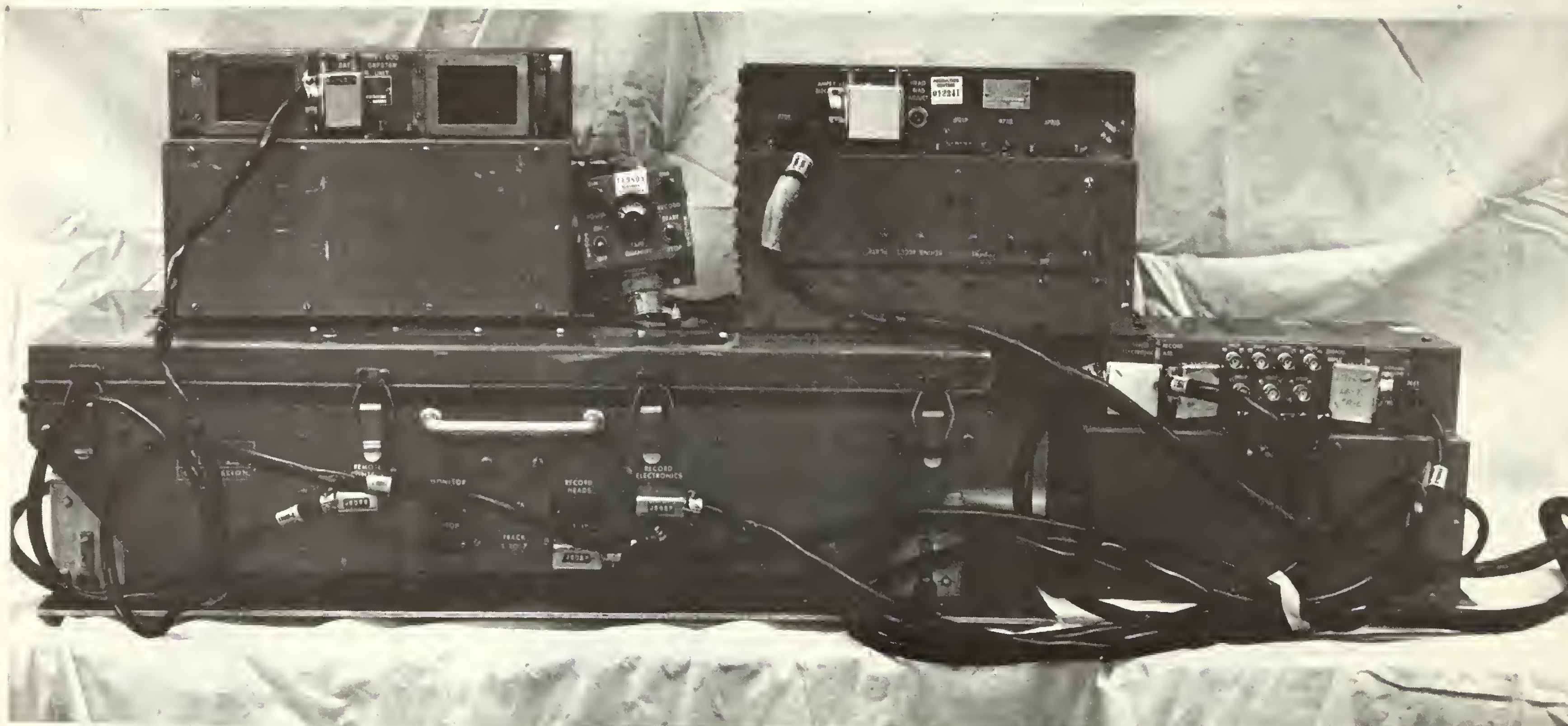

FIGURE 2
AMPEX SERIES 800 MAGNETIC TAPE RECORDER

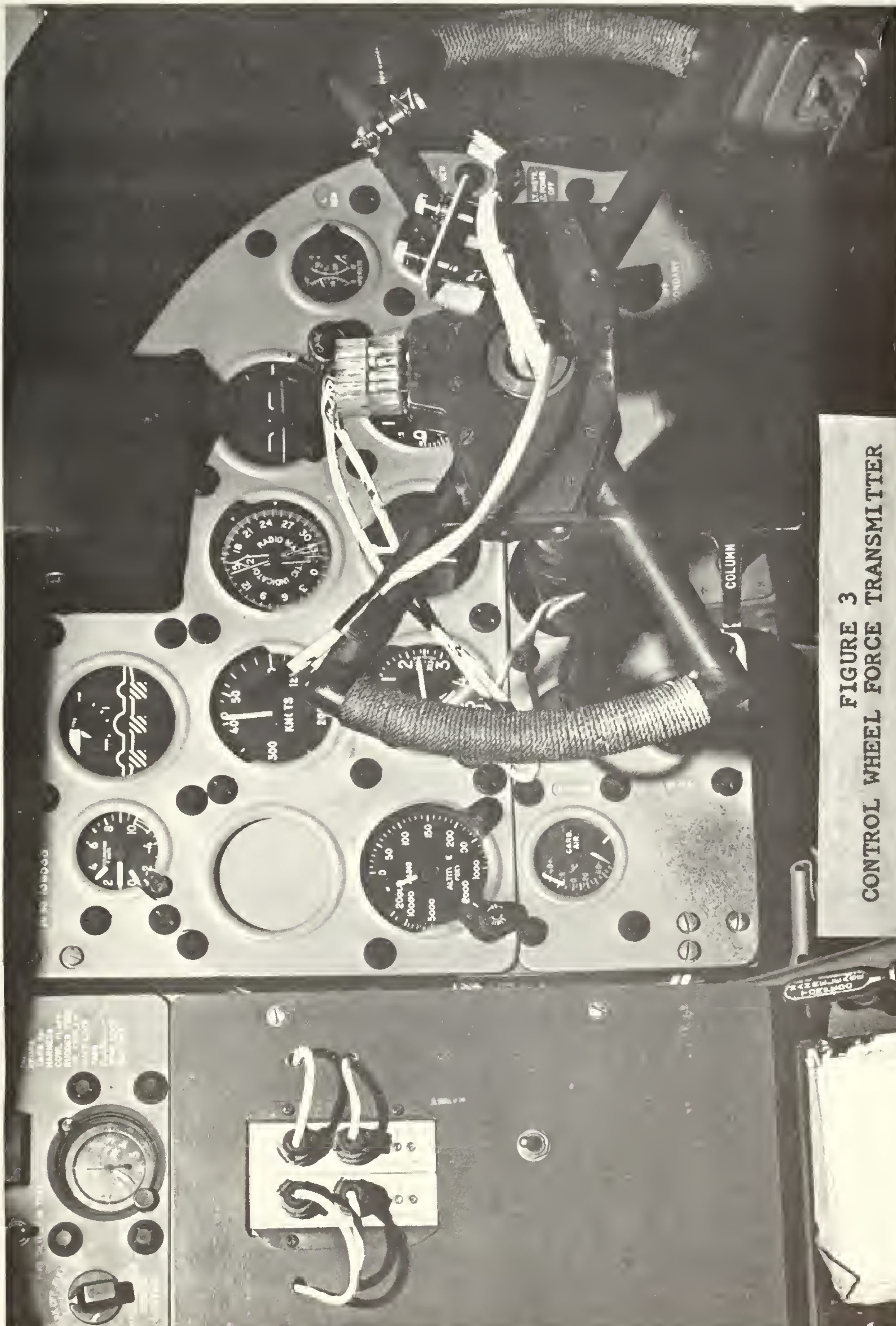

FIGURE 3
CONTROL WHEEL FORCE TRANSMITTER

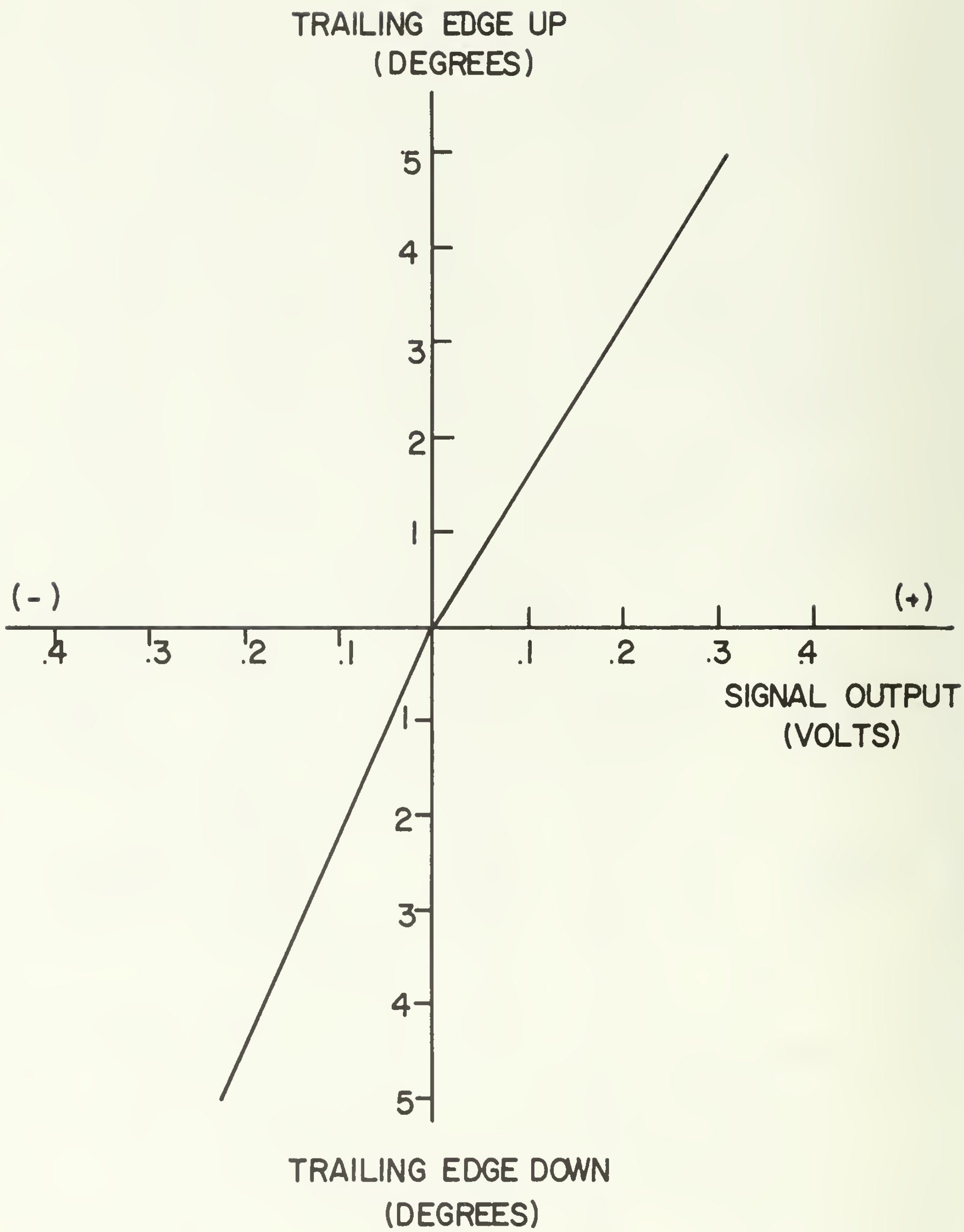

FIGURE 4. ELEVATOR POSITION CALIBRATION

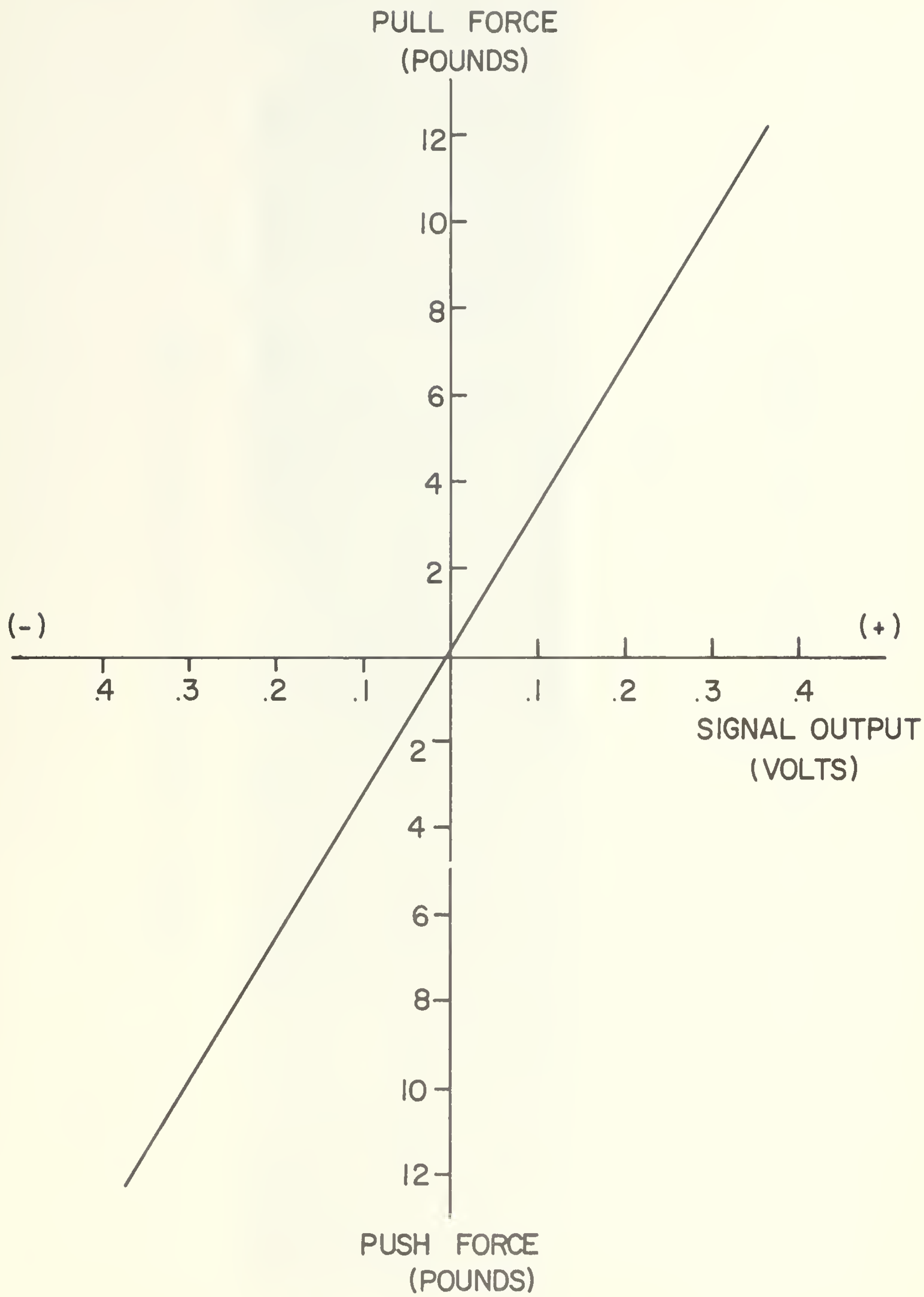

FIGURE 5. CONTROL WHEEL ELEVATOR FORCE CALIBRATION

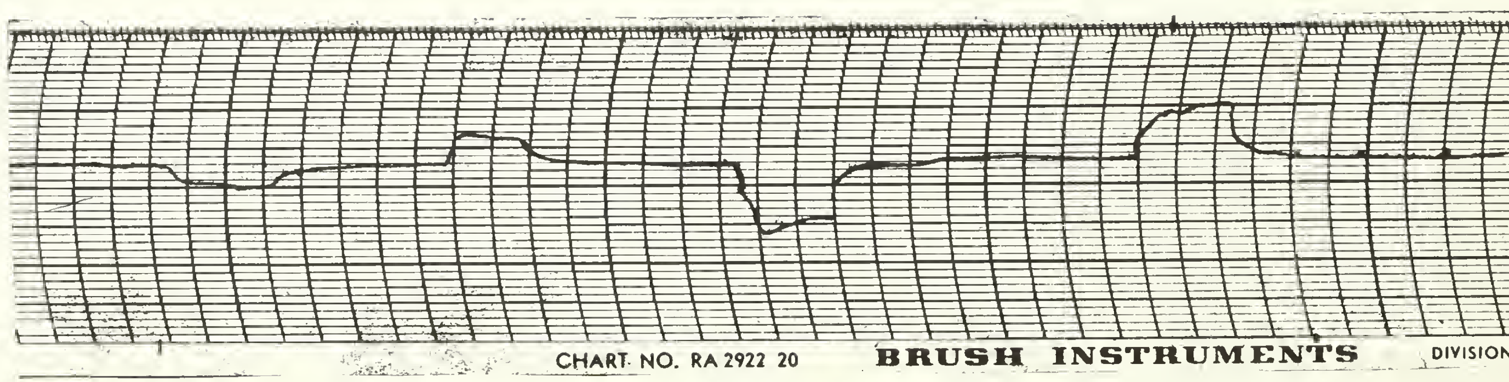

FIGURE 6. SAMPLE GRAPHICAL DISPLAY OF DATA

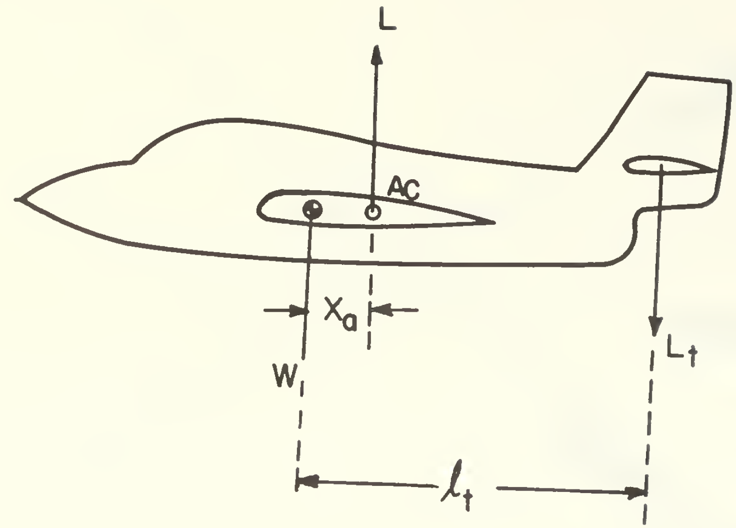

FIGURE 7. FORCES ON AIRCRAFT IN FLIGHT

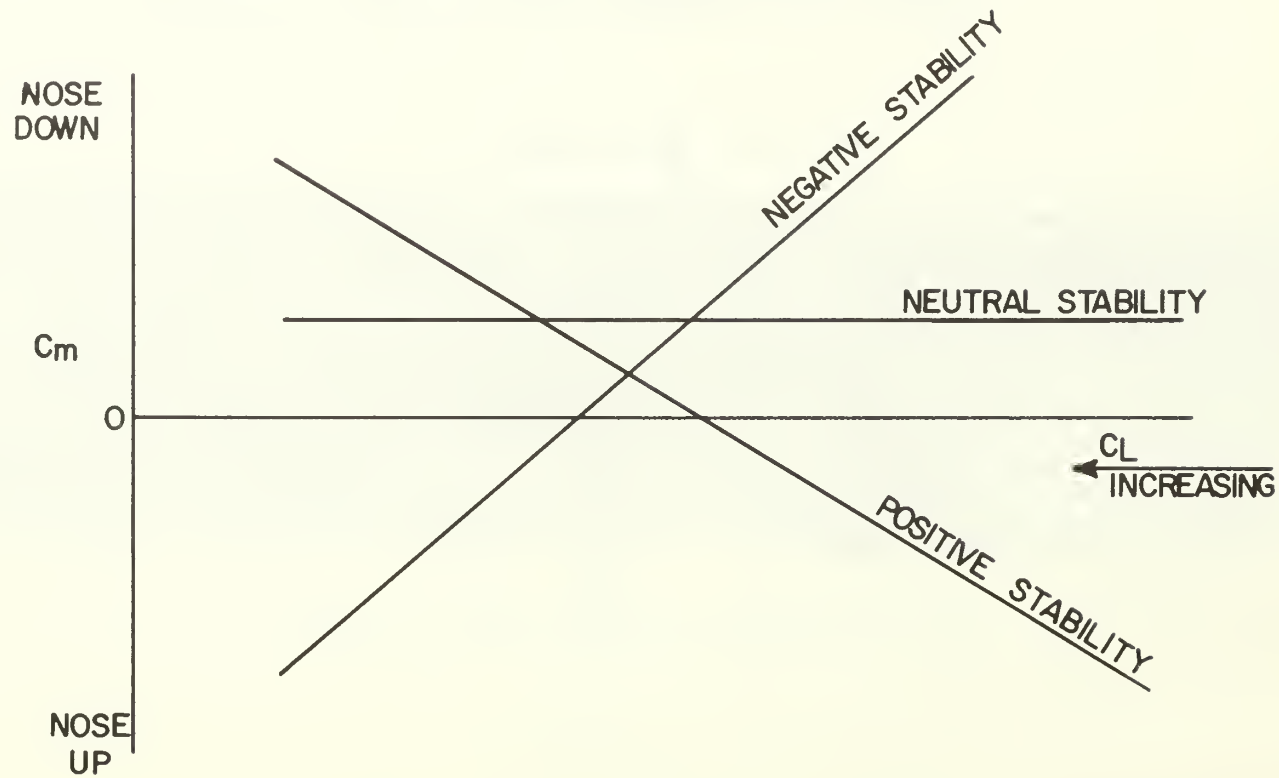

FIGURE 8. SAMPLE C_m VERSUS C_L FOR STABILITY

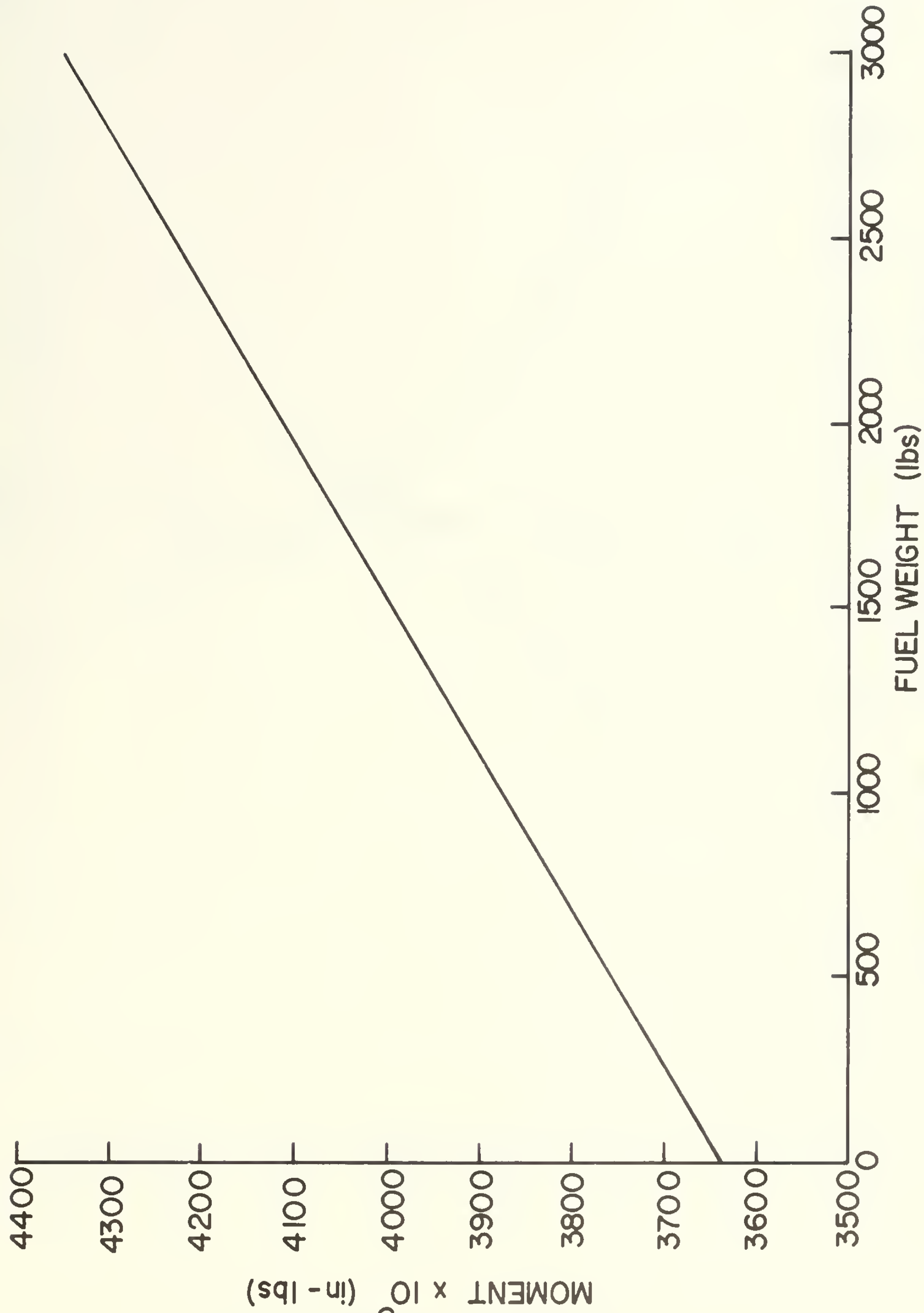

FIGURE 9. MOMENT VERSUS FUEL WEIGHT

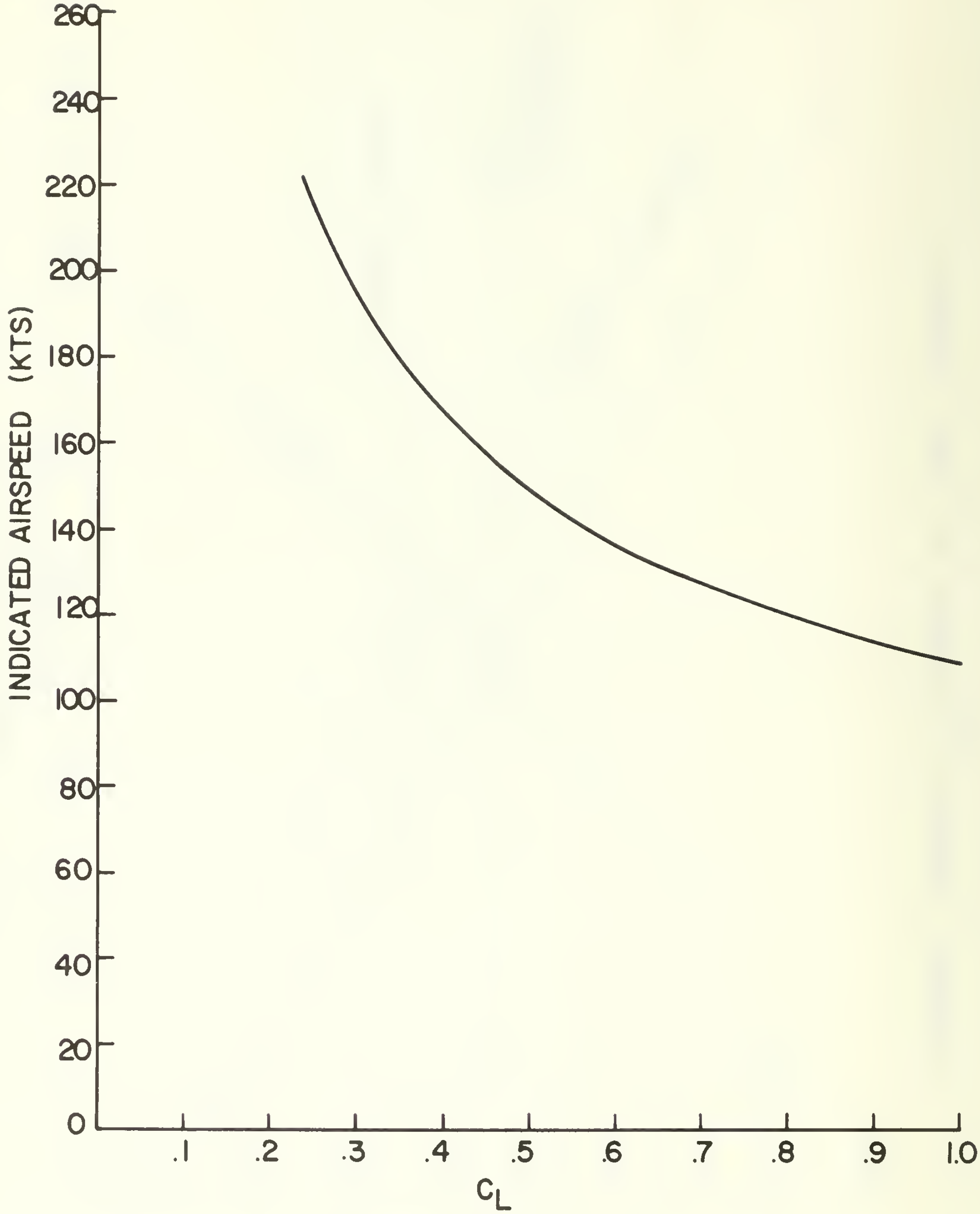

FIGURE 10. INDICATED AIRSPEED VERSUS C_L

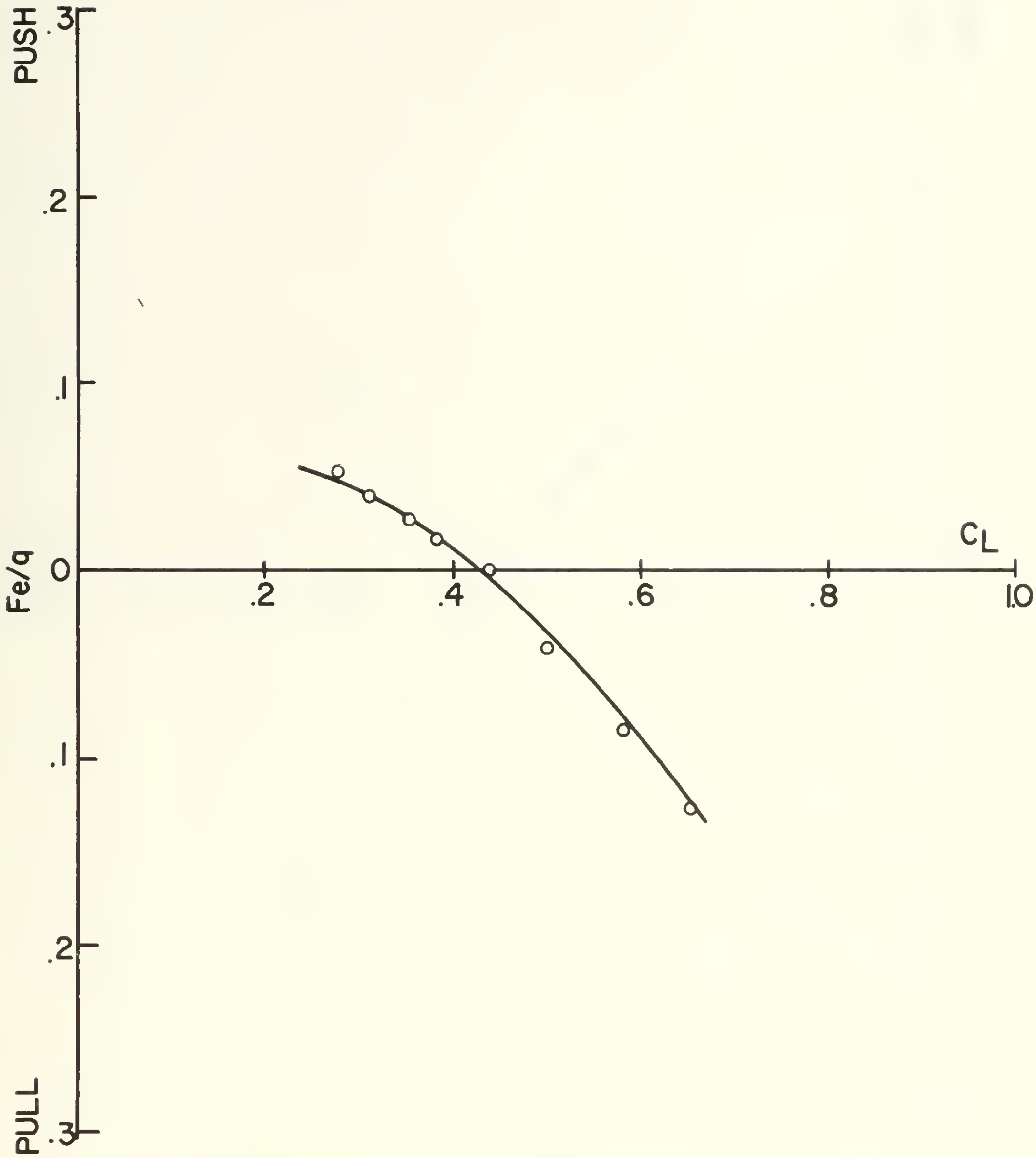

FIGURE 11. Fe /q VERSUS C_L FOR C.G.=24.9% MAC

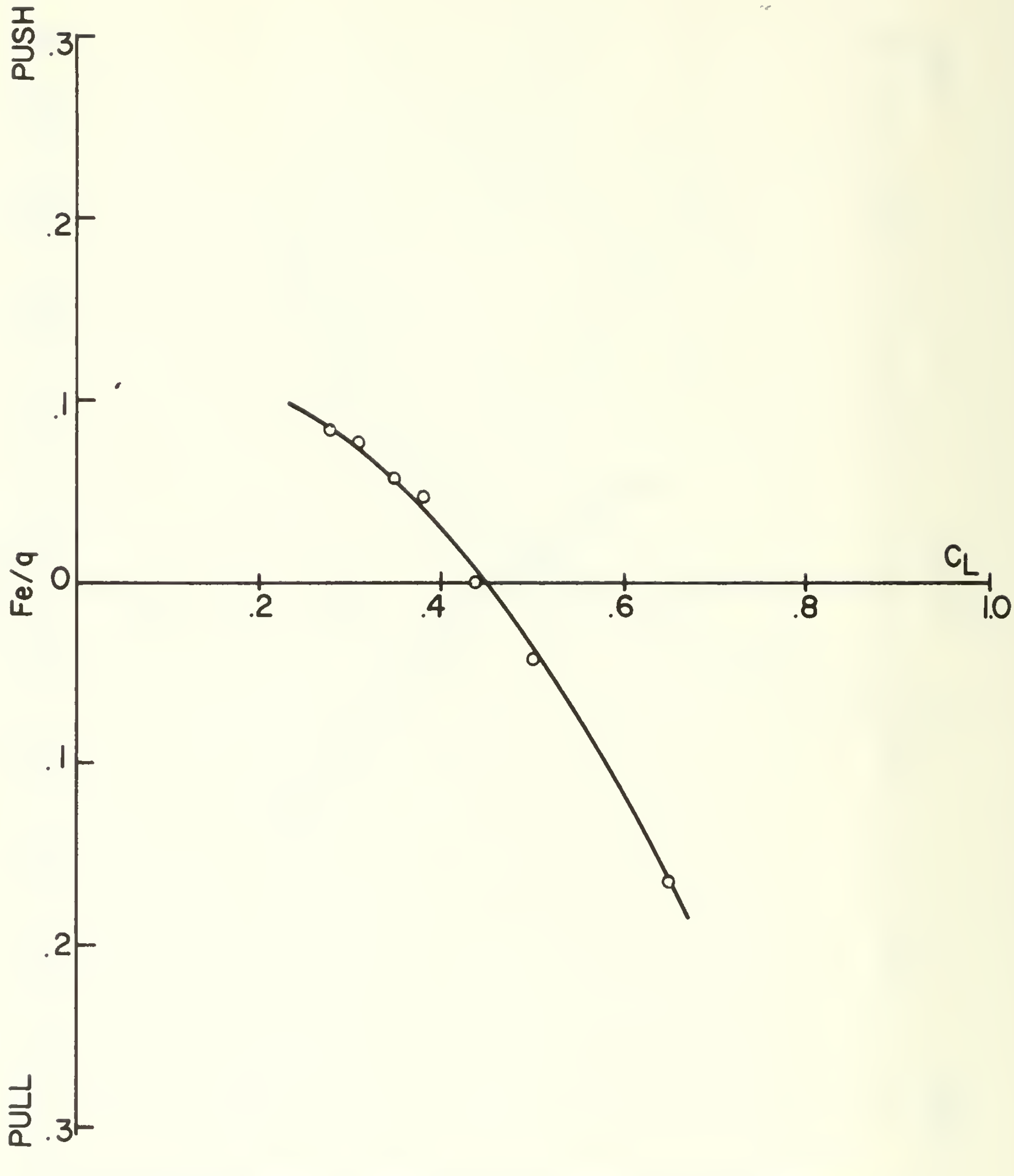

FIGURE 12. Fe/q VERSUS C_L FOR C.G.=22.9% MAC

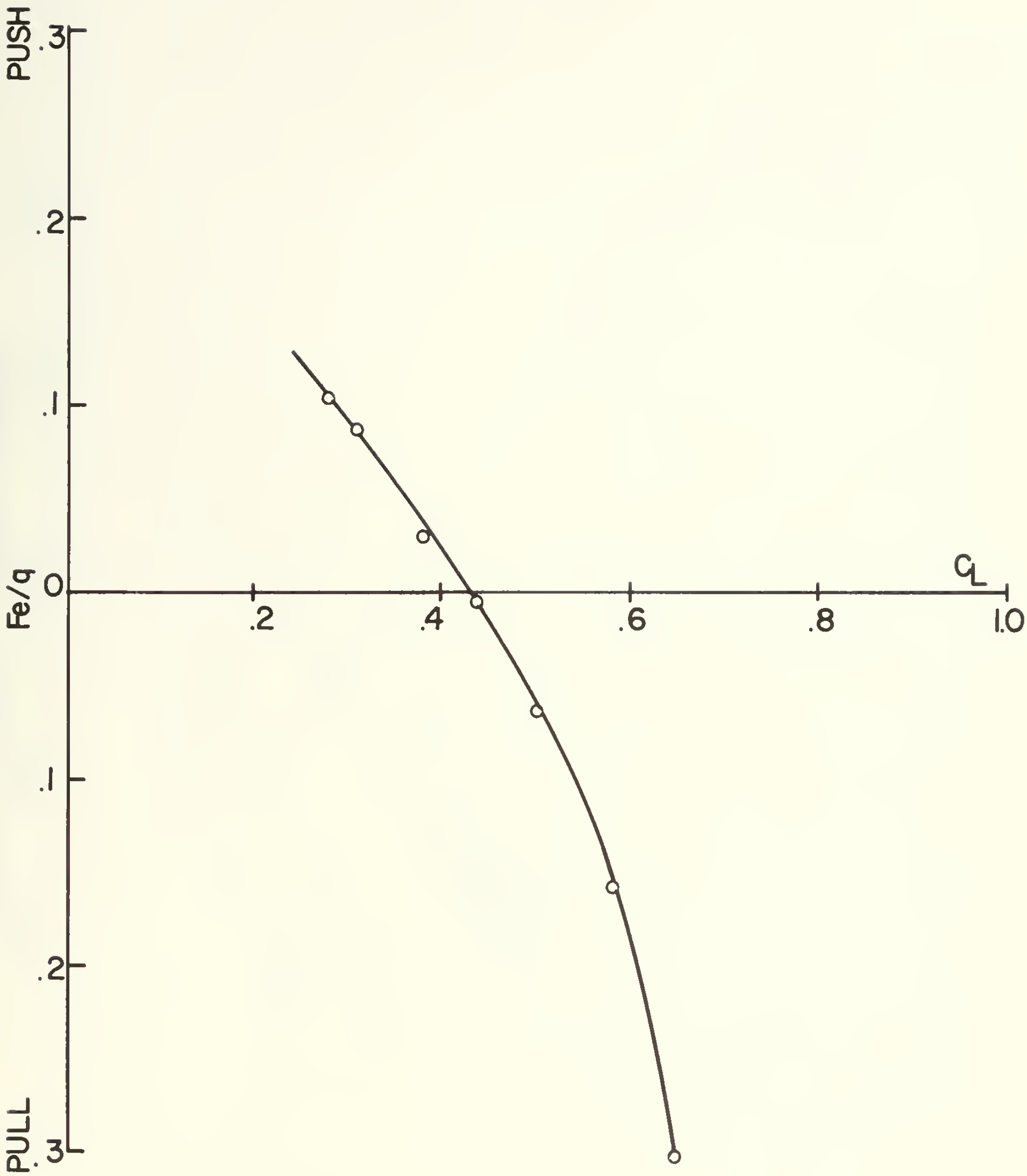

FIGURE 13. Fe/q VERSUS C_L FOR C.G. = 21.3% MAC

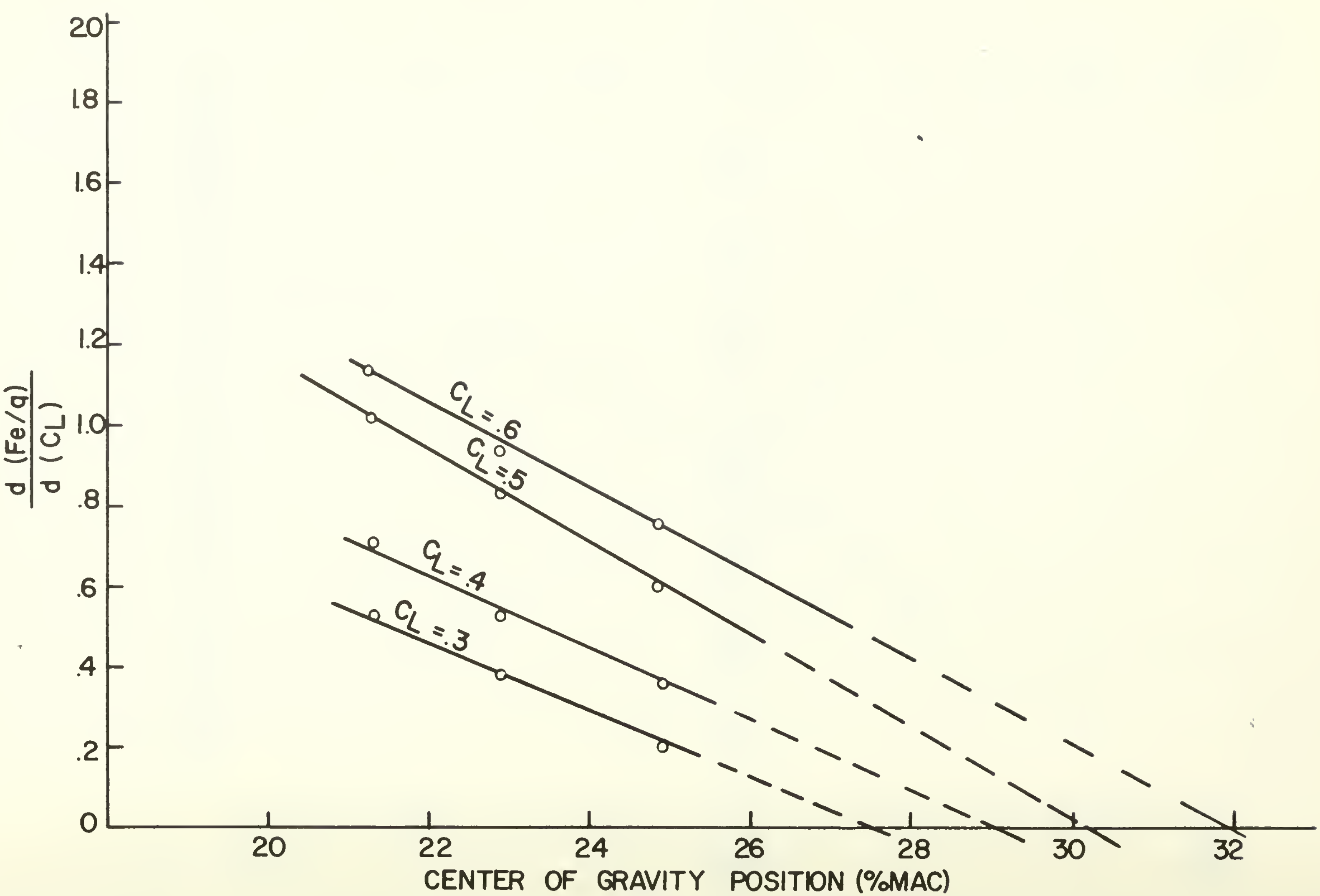

FIGURE 14. $\frac{d(Fe/q)}{d(C_L)}$ VERSUS CENTER OF GRAVITY POSITION

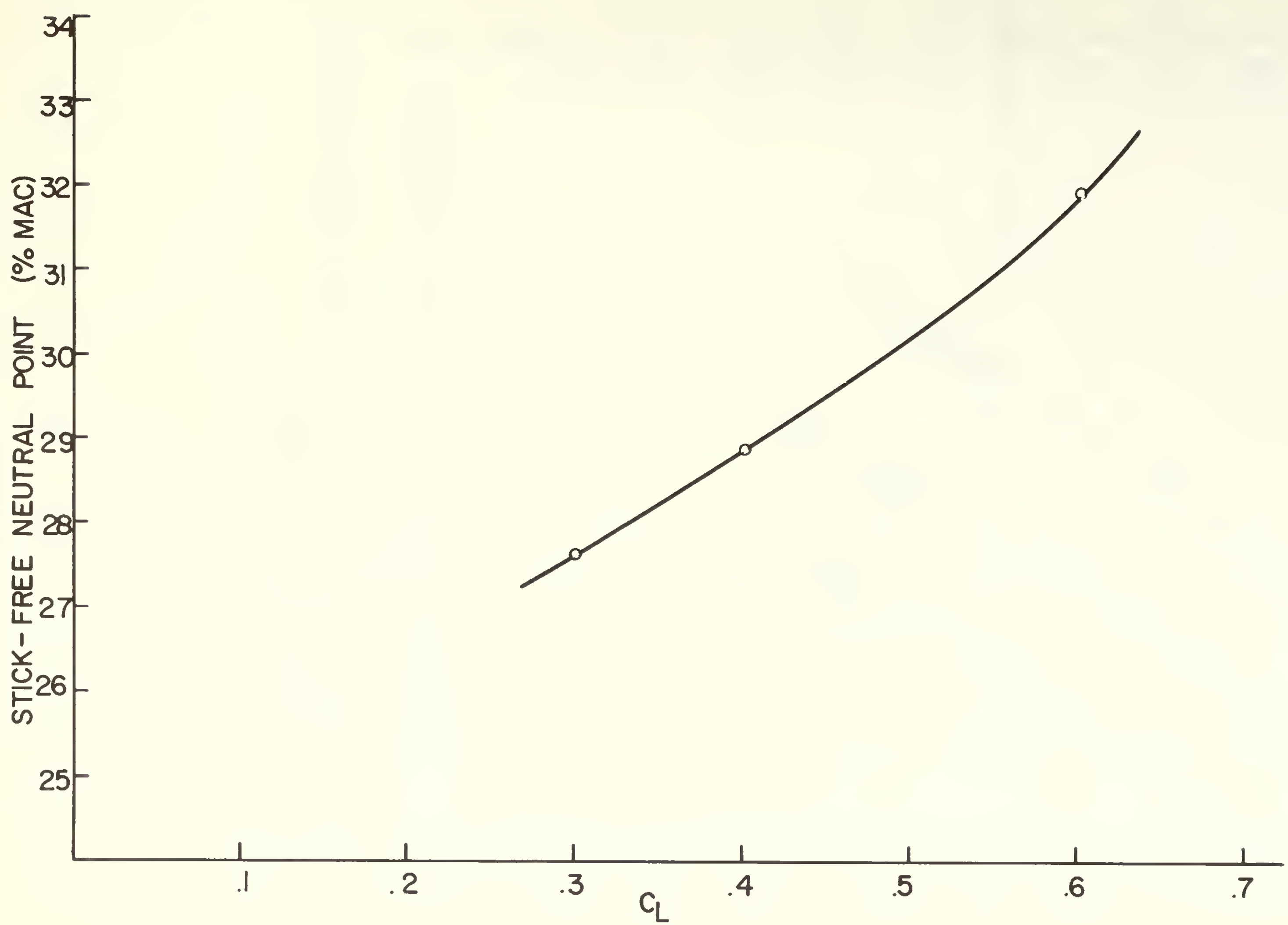

FIGURE 15. STICK-FREE NEUTRAL POINT

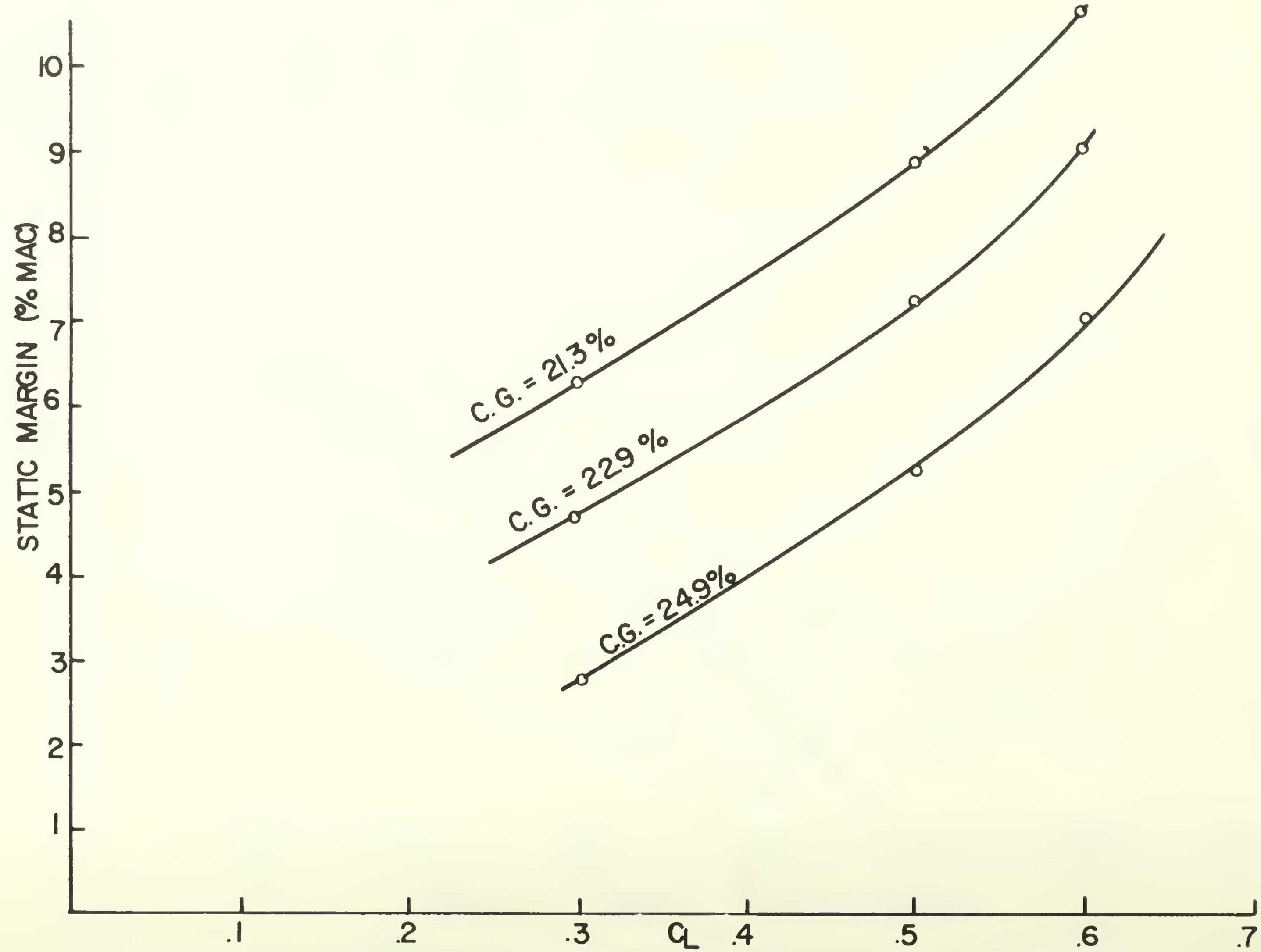

FIGURE 16. STICK-FREE STATIC MARGIN

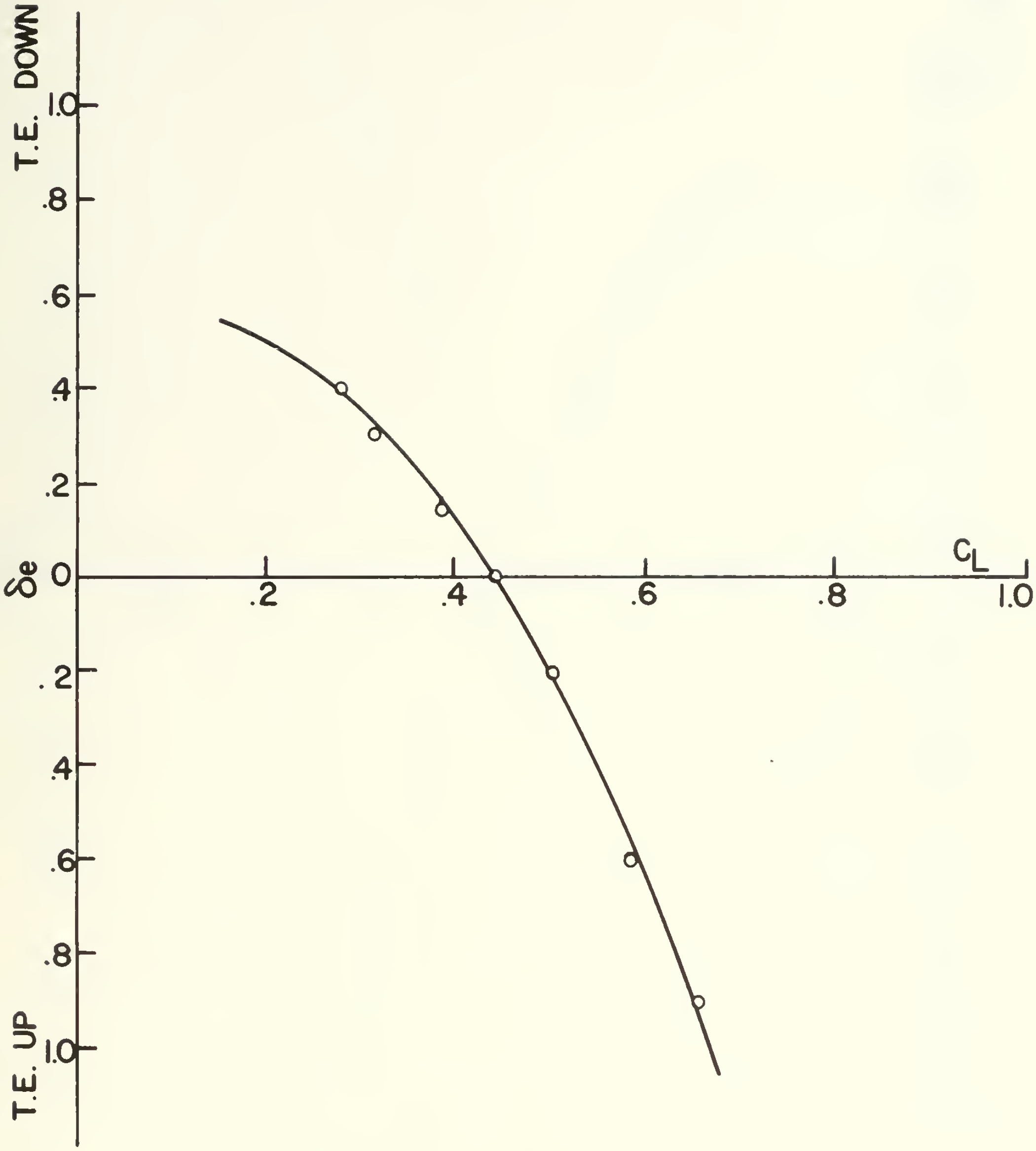

FIGURE 17. ELEVATOR DEFLECTION VERSUS C_L FOR C.G. = 24.9% MAC

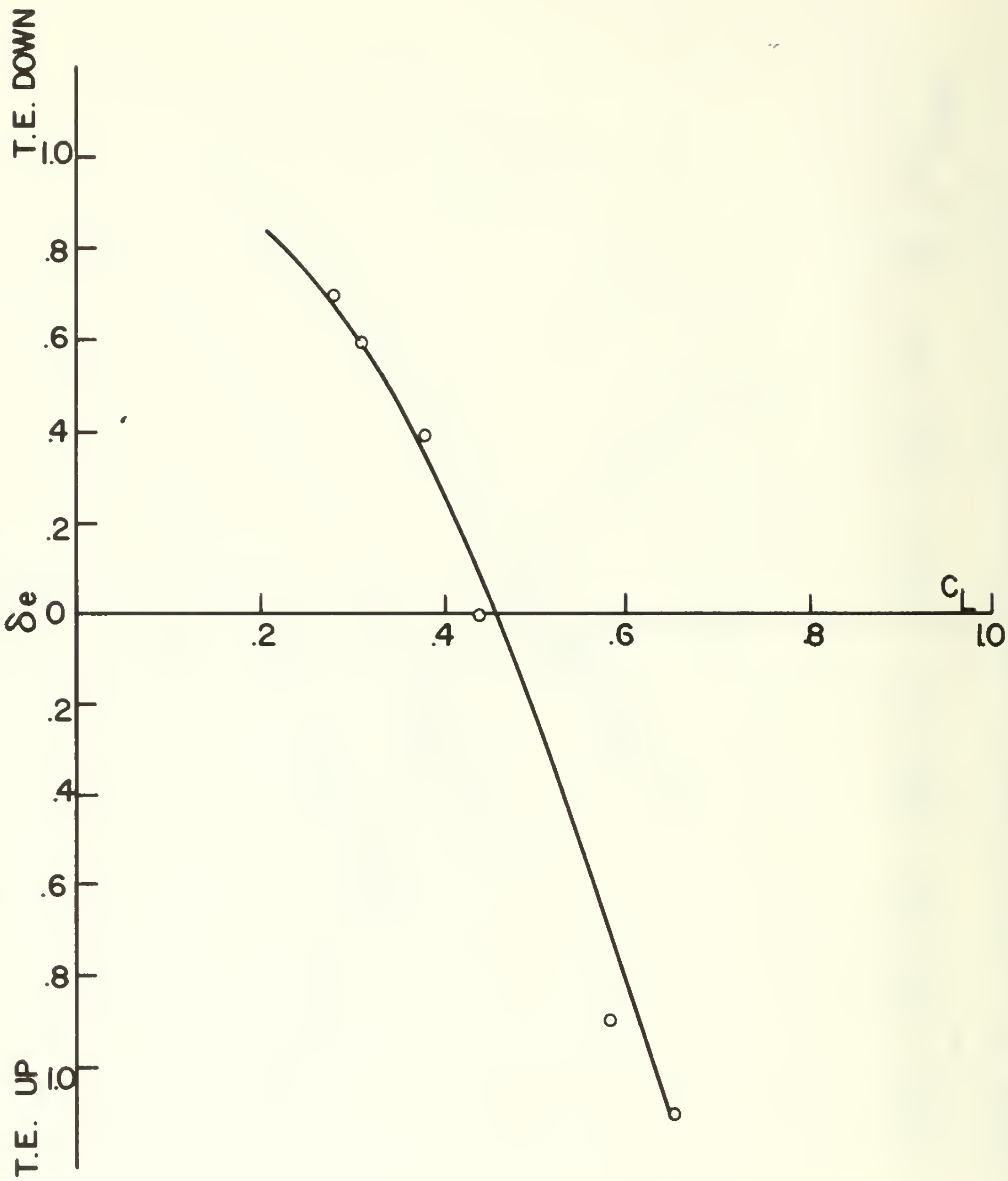

FIGURE 18. ELEVATOR DEFLECTION VERSUS C_L FOR C.G. = 22.9 % MAC

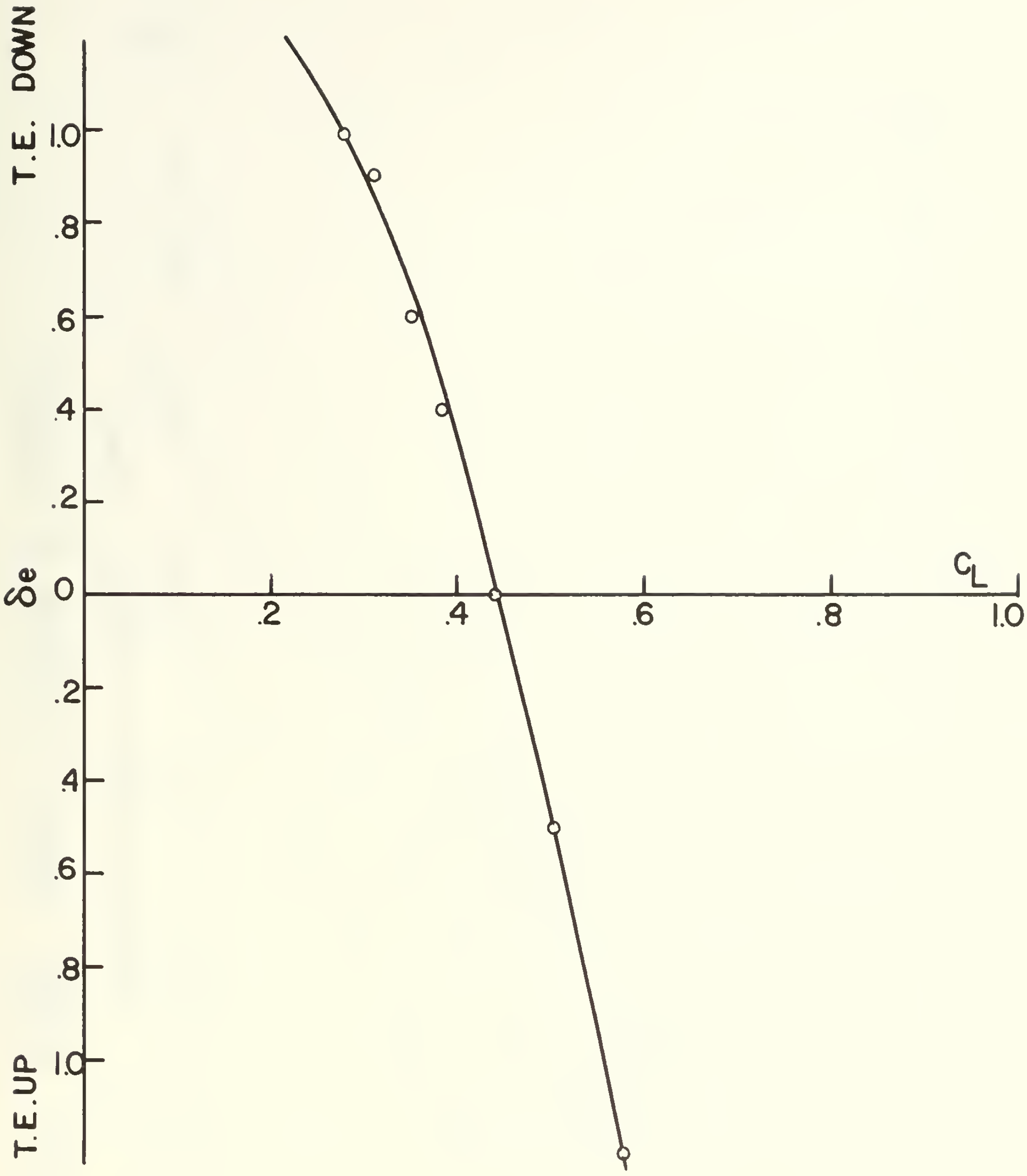

FIGURE 19. ELEVATOR DEFLECTION VERSUS C_L FOR C. G. = 21.3% MAC

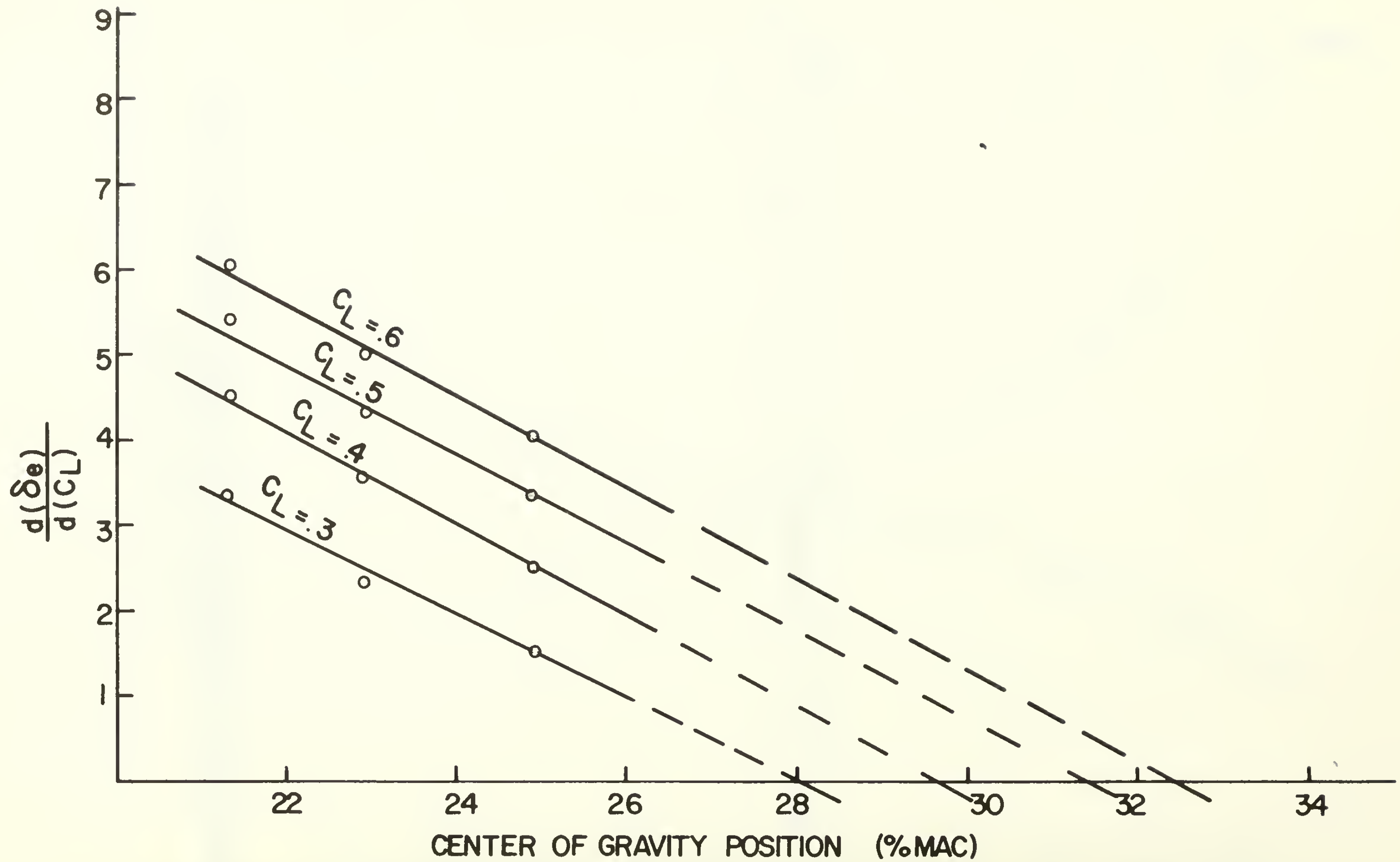

FIGURE 20. $\frac{d(\delta e)}{d(C_L)}$ VERSUS CENTER OF GRAVITY

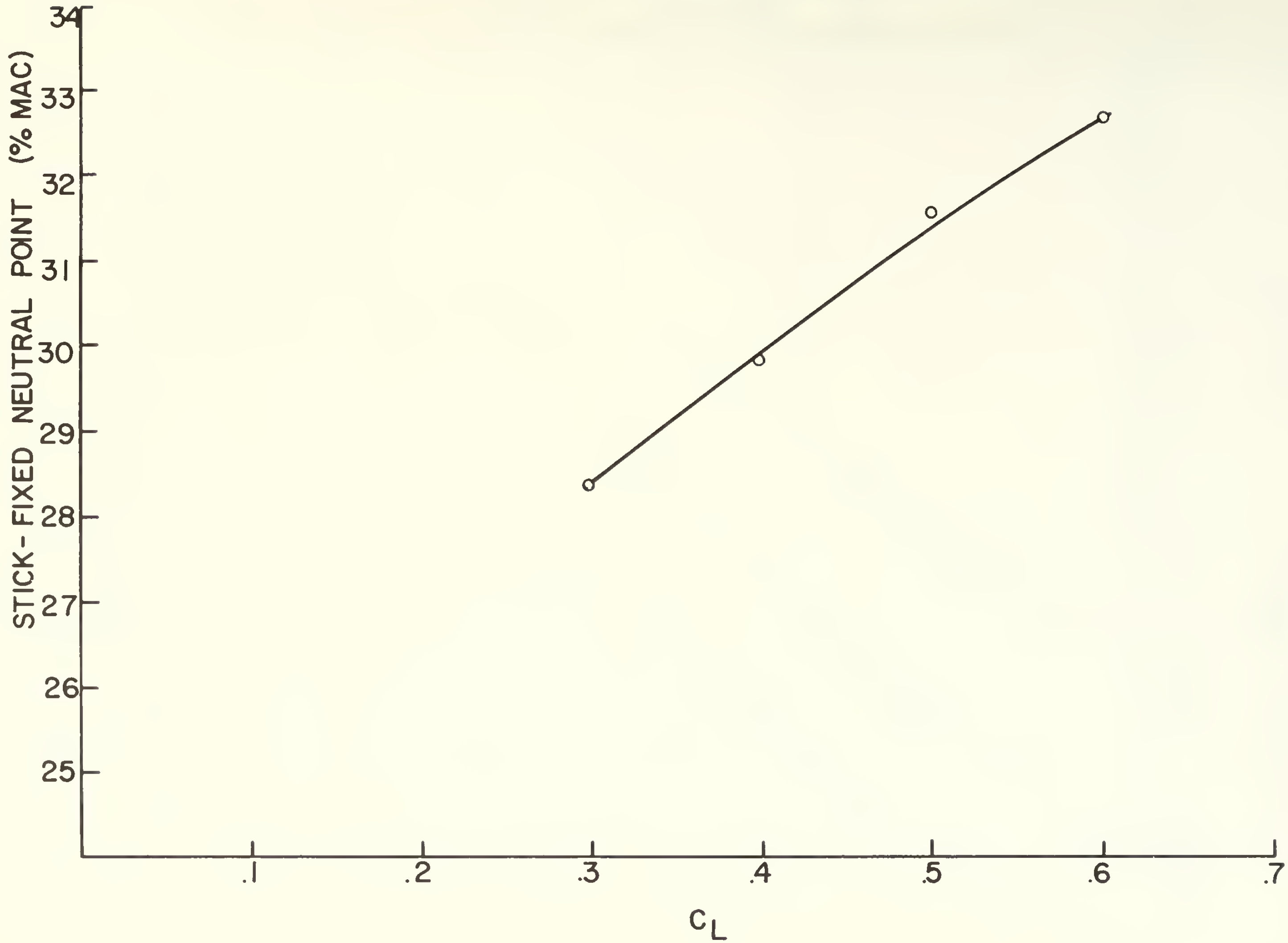

FIGURE 21. STICK - FIXED NEUTRAL POINT

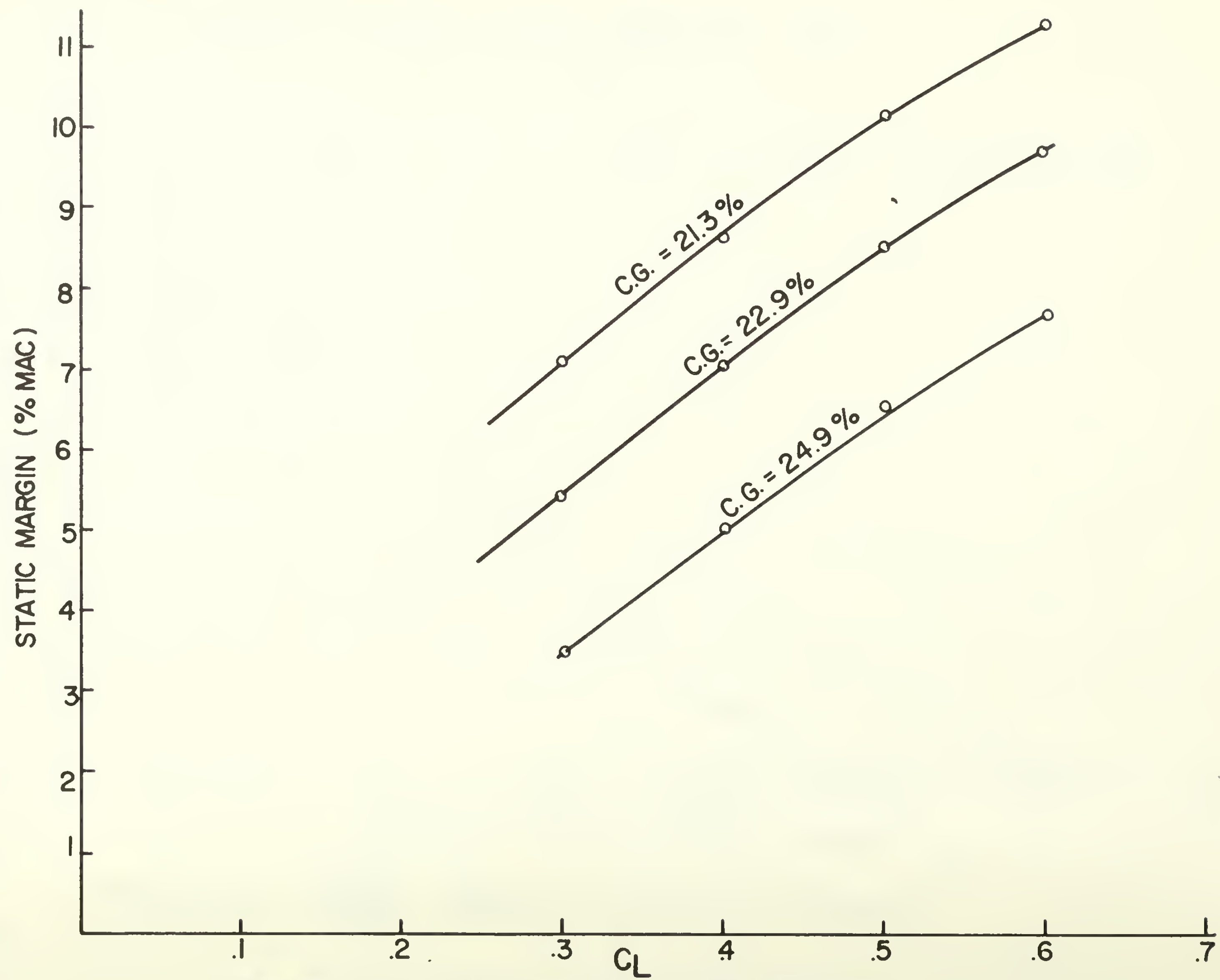

FIGURE 22. STICK-FIXED STATIC MARGIN

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ABSTRACT

An evaluation of the longitudinal static stability of a Navy US-2A aircraft was made. The aircraft was equipped with a Data Acquisition System capable of sensing and recording aerodynamic data other than that given by the standard cockpit instruments. Recorded data were displayed graphically and the Neutral Points for stick-free stability and stick-fixed stability were determined. Stick-free static margin and stick-fixed static margin were determined and evaluated.

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Stability						

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